Spacecraft and mission design for the Precision Optical INTerferometer in Space (POINTS)

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ABSTRACT

This paper describes key features of the instrument, spacecraft, and mission design for the Precision Optical INTerferometer in Space (POINTS), which have evolved through studies at the Jet Propulsion Laboratory during the last few years. This paper complements others in this volume that describe the POINTS instrument, its scientific objectives, and specific engineering concerns. Design of the flight-system configuration has been driven by several considerations. Since the most ambitious science goals require access to a large portion of the sky most of the time, minimal systematic errors, and a 10-year mission life, a high Earth orbit (higher than 50,000 km) is preferred; the nominal has been taken to be a circular orbit of 100,000-km radius. In order to provide a very uniform thermal environment for the instrument, a solar shield supporting an array of solar cells is mounted on a boom and gimballed along two axes so as to remain pointed at the Sun and to provide constant shade for the entire spacecraft. Silicon cells covering about 85% of the roughly 4.8-m-diameter shield and operating at about 100 deg C could supply sufficient power for a 10-year mission life. A unibody design was selected in which the instrument and spacecraft bus are solidly attached to form a single rigid body. Full pointing freedom for the instrument is provided by articulation of the solar shield about two axes plus roll of the entire spacecraft around the Sun direction. With the high orbit and Sun facing, geometrically simple spacecraft configuration, the effects of solar radiation pressure the only significant external disturbance to spacecraft acceleration can be modeled accurately enough to guarantee no compromise in the accurate velocity determination needed to correct astrometric measurements for stellar aberration.

The principal mechanisms for POINTS include that which accomplishes the few-degree articulation between the two orthogonal interferometers, the two gimbal actuators between the solar shield and the spacecraft, and the reaction wheels used for spacecraft attitude control. The designs for these are discussed. Cold-gas thrusters have been included for spacecraft angular momentum management, although solar-pressure torque also could be used for axes other than the Sun line. A layered thermal control system provides millidegree temperature stability over periods of up to several hours for critical instrument optical elements.

Systems issues discussed here include launch sequence, power requirements, spacecraft velocity determination, and telemetry and data storage. The proposed launch vehicle is an Atlas IIAS with a Centaur upper stage, plus a solid-fuel rocket motor for orbit circularization. Spacecraft velocity determination is accomplished by tracking a GPS-like beacon on the spacecraft with the existing ground-based network of GPS receiving antennas. This method is predicted to be easily capable of determining velocity to an accuracy of 0.6 mm/s, sufficient to correct for stellar aberration to 0.4 microarcseconds (μas). Telemetry requirements can be met with S- or X-band downlinks to existing ground-based antennas, and data storage on the spacecraft can be handled with a 500-megabyte solid-state memory such as that baselined for the upcoming Cassini mission.

1. INTRODUCTION

Since 1989, NASA's Jet Propulsion Laboratory (JPL) and the Smithsonian Astrophysical Observatory (SAO) have been collaborating on the design of and technology development for an orbiting astrometric interferometer that would be capable of establishing a global optical reference frame and measuring angular positions of roughly 1000 additional stars and astrophysical objects with a nominal measurement precision of five microarcseconds (5 μas), in a mission lasting several years. To serve as the basis for a definitive search for giant planets around other stars, observations would have to be repeated several times per year for as long as 10 years.[1] The baseline instrument concept has been SAO's POINTS (Precision Optical INTerferometer in Space).[2] The POINTS concept originated in the mid-1970s as a means to perform a stringent light-deflection test of general relativity near the solar limb.[3] Since then, it has evolved and is fast acquiring engineering maturity as an instrument capable of providing conclusive evidence of the prevalence of planetary systems and of performing a variety of fundamental astrophysical measurements.

During the last three years, JPL has undertaken design and analysis of the spacecraft and instrument configurations for POINTS and examination of critical operational issues for a POINTS mission.[4] This paper offers a brief overview of some of that work. Section 2 describes the current spacecraft flight-system configuration, including overall design drivers and trade-offs. Section 3 describes the instrument configuration, including the housing, optical benches, and the mechanism by which one interferometer is articulated relative to the other. Section 4 describes attitude and articulation control, including overall requirements, actuator design, and spacecraft angular momentum management. Section 5 describes thermal control for the spacecraft and instrument, with emphasis on the strategy for providing millidegree temperature control for critical optical elements in the instrument. Section 6 discusses several systems issues, including the launch sequence, power requirements, spacecraft velocity determination, and telemetry and data storage.

2. SPACECRAFT FLIGHT-SYSTEM CONFIGURATION

The ambitious science goals for POINTS[2,5] impose a variety of requirements on the instrument, spacecraft, and mission. These include access to a large fraction of the sky most of the time; a low-disturbance environment for the spacecraft and instrument; telemetry and data storage consistent with a few hundred megabytes of data per day; precise velocity determination to correct for stellar aberration at the sub-microarcsecond level; and a mission life of 5 to 10 years. NASA's desires that such a mission cost no more than about $400 million (exclusive of launch vehicle) and require only an intermediate-size launch vehicle impose constraints on spacecraft and instrument mass, size, complexity, and technology maturity. These requirements and constraints lead to a number of design drivers and trade-offs, some of which are summarized below.

2.1 Orbit choice

The demands for target accessibility, a low-disturbance, thermally quiet environment, and a long mission life are best met with a heliocentric orbit or a high Earth orbit. The long mission life and large quantity of data make a heliocentric orbit problematic, since the spacecraft would drift as much as one AU from Earth over a 10-year mission. A high Earth orbit poses different challenges than does a low orbit for spacecraft angular-momentum management and telemetry, and it imposes more severe constraints on instrument mass. Furthermore, since the required correction to astrometric measurements for stellar aberration scales with the sine of the angular separation between target and reference stars, a wide-angle instrument such as POINTS requires more accurate orbit determination than would a narrow-angle instrument; thus, velocity determination accuracy must be better than 1.5 mm/s for sub-microarcsecond correction. Since each of these challenges is met with the current POINTS design (as described below), the benefits of a high Earth orbit can be exploited. Although any orbit above about 50,000 km would suffice to ensure infrequent eclipses and low contamination hazard, the nominal choice, which allows for conservative design margins, is a circular orbit of 100,000-km radius.

2.2 Thermally and vibrationally quiet instrument environment

In order to achieve maximal scientific return for a spacecraft of a particular size and cost, the design of a high-accuracy astrometric instrument calls for high dimensional stability for certain optical elements, which in turn requires a low-disturbance environment. For the small POINTS instrument to meet its most ambitious science goals, one of the most demanding of these requirements is local temperature stability in the range of a few millidegrees for several optical elements, for periods of up to several hours. This is facilitated in part by the fact that, in the POINTS design, these optical elements are small and buried deep within the instrument. To further aid in achieving this stability, it was decided to provide permanent shade for the entire spacecraft (instrument plus bus) with a solar shield, supporting an array of solar cells, that is mounted on an extended boom. To guarantee full shade at all times for the spacecraft and boom, the shield radius R (the shield is assumed circular here) and boom length L must satisfy

$$R \cos g - L \sin g > S \,, \tag{1}$$

where S is the maximum projected dimension (radius) of the spacecraft perpendicular to the boom (parallel to the solar shield), and g is the desired clearance angle from normal solar incidence. The angle g is on the order of a few degrees; it includes allowances for spacecraft attitude variation, penumbra convergence, and safety margin. Referring to the nominal configuration shown in Figure 1a, S is the sum of the boom width d at the "elbow" between the two solar-shield gimbals and the root-sum-square of the instrument housing radius and height. The housing radius is limited to about 1.8 m by the Atlas IIAS[6] fairing, and the housing height is about 1 m. (See Section 3.1.) For the nominal configuration, $R = 2.4$ m and $L = 4.35$ m; thus, if g is allowed to be as large as 2 deg, the boom elbow width d cannot exceed about 18 cm, whereas d could be as large as 26 cm if g does not exceed 1 deg. This constraint has important consequences for the vibrational characteristics of the spacecraft, since boom bending and twisting frequencies scale roughly as $w^{1/2}d^{3/2}/(L^{3/2}R)$ and $w^{1/2}d^{3/2}/(L^{1/2}R^2)$, respectively, for a uniform hollow boom with wall thickness w, and shield bending frequencies scale roughly as h/R^2 for a shield of thickness h.

Larger clearance angles from normal solar incidence would require that the solar shield be larger, in order to maintain shade for the instrument for all interferometer orientations. Note, however, that $R > 2.4$ m would require deployment of an extension to the nominal shield, since $R = 2.4$ m corresponds to the maximum size that can be accommodated in the launch configuration (see Section 2.4). The alternatives to increasing solar shield size, in order to allow larger clearance angles and still maintain shade for the spacecraft for all interferometer orientations, are unacceptable; a shorter boom would compromise target accessibility (see Section 2.3), and a thinner boom would compromise stiffness. In general, the choices for boom length L, boom width d, and solar-shield radius R must satisfy

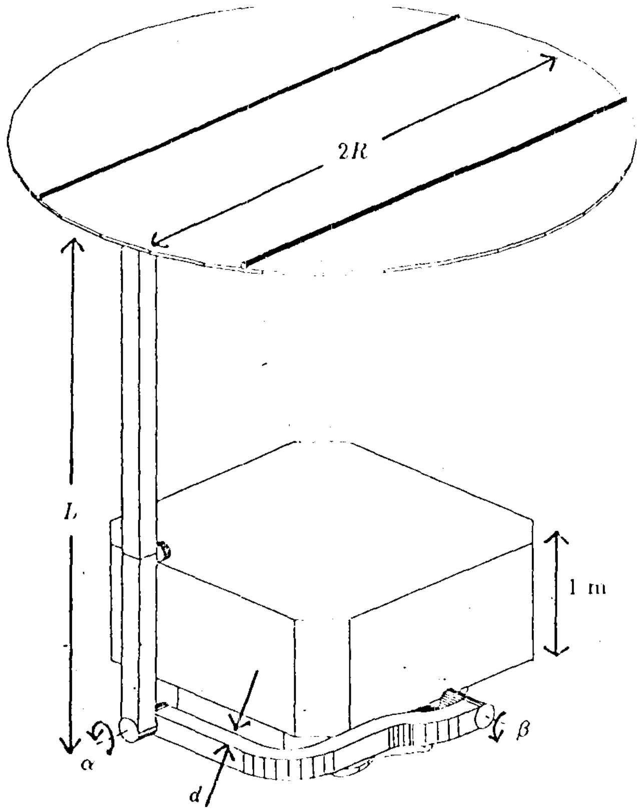

Fig. 1a. POINTS spacecraft, fully deployed. Solar shield radius R, support boom length L, and boom elbow diameter d are defined in eq. (1) and surrounding discussion.

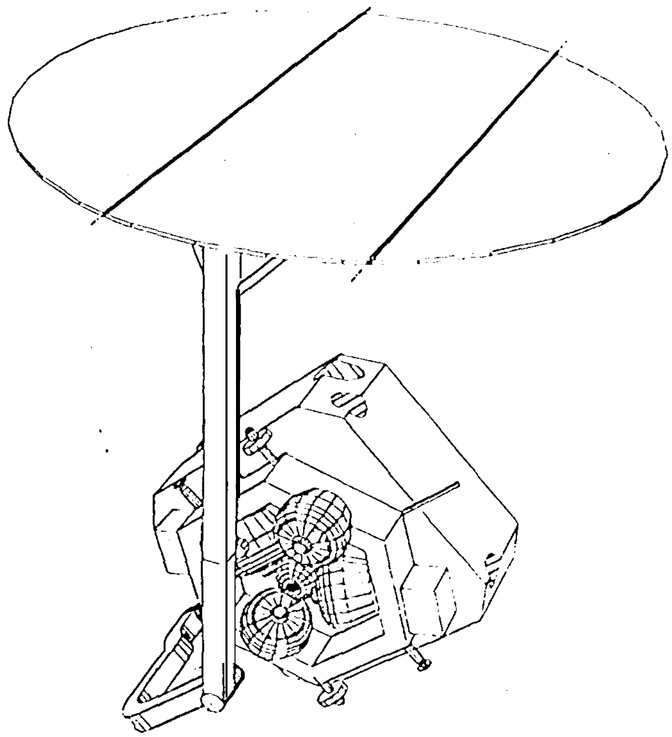

Fig. 1b. POINTS spacecraft, fully deployed, during a slew. Spacecraft bus is shown attached to the instrument housing.

the combined requirements on shade for the spacecraft, target accessibility, and overall vibrational characteristics, as well as the constraints imposed by launch-vehicle size.

To aid in maintaining a vibrationally quiet environment for the instrument, it was decided to attach the spacecraft bus rigidly to the instrument housing, and to articulate the solar shield relative to the entire spacecraft *via* two single-axis gimbals, which go through the spacecraft center of mass (the α and β axes identified in Figure 1 a), Earlier concepts that involved articulation of the instrument relative to the spacecraft bus were rejected in favor of the current unibody design in which only the solar shield is gimballed. This design minimizes launch loads through the gimbal actuators as well as structural flexibility between the instrument and reaction wheels, and is favored by findings that internal vibrational disturbances (primarily from reaction wheels) can be isolated effectively at their source. It also is thought to minimize the mass, complexity, and cost associated with the gimbal mechanisms. (For further discussion, see Section 2.4.)

Simpler configurations involving body-mounted or single-axis-gimballed solar panels covering multiple sides of the spacecraft were rejected in favor of the two-axis-gimballed, Sun-oriented shield because the former suffered from several disadvantages. First, both pointing capability and target accessibility would be more limited (see Section 2.3 below). Second, the instrument thermal environment would be less uniform, and thermal creak and flap of the arrays could degrade pointing stability. Third, since only 30% to 50% of the solar cells would be providing power at any time, the total array mass would be greater by as much as 200 kg. A further consideration in favor of the geometrically simple, Sun-facing configuration is that it permits accurate modeling (to better than 5%) of the effects of solar radiation pressure, the only significant external source of error in modeling the spacecraft acceleration.

2.3 Maximum target accessibility

To maximize science return, a large fraction of the sky should be accessible to the instrument at all times. This is especially true in a search for other planetary systems, where lack of access to the sky near the Sun would impose on the data a window with an annual period. Thus, one would like to minimize the Sun-avoidance angle A (the closest either interferometer can look toward the Sun direction), without exposing the spacecraft or boom to the Sun. An approximate measure of target accessibility is provided by $V = \cos^2(A/2)$, the fractional solid angle of sky accessible to either interferometer. For normal solar incidence, the avoidance angle A_0 is related to boom length L, shield radius R, and perpendicular distance T between the spacecraft center-of-mass and the optical axis of the telescope looking closest to the Sun by

$$L \sin A_0 \quad R \cos A_0 = T . \tag{2}$$

The actual avoidance angle A may be smaller than A_0 to the extent that deviations from normal incidence do not expose the spacecraft or boom to the Sun; thus, $V \geq V_0 \equiv \cos^2(A_0/2)$. For a given avoidance angle A_0, clearance angle g from normal solar incidence, and boom elbow width d, the combined constraints of equations (1) and (2) produce a family of allowable values for R and L. For the nominal configuration, with $T = 1.125$ m, $d = 18$ cm, and $g = 2$ deg, $A_0 \approx 42$ deg, making at least 87% of the sky accessible at any time to each interferometer.

The fraction of sky accessible to each interferometer at any time (the quantity V_0) can be increased slightly from that provided by the nominal configuration, but it requires considerable increase in boom length L, which in turn requires an increase in boom width d in order to maintain stiffness. For example, the angle A_0 can be reduced to about 40 deg to give $V_0 \approx 0.88$ with $d = 24$ cm, $L = 4.7$ m, and $R = 2.47$ m (assuming $g = 2$ deg). Furthermore, these choices would result in higher fundamental vibrational frequencies than with the nominal configuration. Note, however, that $L = 4.7$ m corresponds approximately to the longest boom (if folded in only one place) that can be accommodated in the launch configuration. Similarly, since the launch-vehicle configuration cannot accommodate a circular solar shield with R greater than about 2.4 m, these choices would require a noncircular shield or additional deployable "umbrella" to provide an effective R of about 2.47 m. (See Section 2.4.)

2.4 Spacecraft configuration

The nominal POINTS spacecraft configuration is shown in Figures 1a–f. Figures 1a and 1b show the spacecraft as it would appear during the mission, with the instrument and solar shield in two of a variety of possible orientations. The nominal configuration has a 4.8-m-diameter circular solar shield and a single, hollow graphite-epoxy boom 4.35 m long, with square cross section 18 cm on a side and walls 3.5 mm thick ($R = 2.4$ m, $L = 4.35$ m, $d = 18$ cm, $w = 3.5$ mm). The gimbals have been arranged so that their axes pass through the spacecraft center of mass, which simplifies shading geometry and reduces excitation of the solar shield and support boom for two of the three axes of spacecraft slewing. The relation of the gimbal axes to the interferometer optical axes (instrument boresights) is such that the support boom need not obscure target stars.

NASTRAN analyses indicate that this configuration can be made stiff enough that all spacecraft vibrational mode frequencies are higher than about 2 Hz, with the lowest modes dominated by boom bending. The frequencies of these modes could be raised with a variety of straightforward modifications (all affordable to the spacecraft mass, size, and cost budgets), the most obvious of which would involve a thicker, stiffer boom and a non-circular solar shield or small deployable umbrella to provide the required additional shade; damping could be improved by coating the boom. The option of a multi-arm support structure to increase stiffness was rejected because of the increased sky obscuration and mass. The requirements on spacecraft vibrational characteristics are driven ultimately by the desired rapid scheduling of science observations — an average of about 3 minutes between the completion of one observation and the start of another, with the average slew between observations being about 25 deg. This rapid scheduling could be met with the nominal configuration, using available passive isolation technology for internal disturbance sources (primarily reaction wheels).

Figure 1c shows the POINTS launch configuration, packaged to fit within the Atlas IIAS shroud. The solar shield is folded in two places to fit in the launch envelope. After launch-vehicle separation, the solar shield (but not the support boom) is deployed and rigidized with mechanical or magnetic latching devices, to provide power during the 17-hour cruise (see Figure 1d). Orbit circularization is accomplished with a Star 37FM[7] solid-fuel rocket motor. Just prior to its ignition, a spin-up solid motor is fired to provide a spin rate of a few RPM; after circularization, a spin-down motor is fired, and the spacecraft returns to a 3 axis-stabilized mode. The spin-up and spin-down thrusters are located in planes that contain the spacecraft center of mass at the times the respective thrusters are fired. (The spacecraft center of mass migrates forward after expenditure of the Star 37FM propellant.) The propulsion module provides structure for integration of the Star 37FM and spin-up and spin-down solid motors, launch support for the lower portion of the stowed solar shield, and structure for linking the spacecraft to the launch-vehicle adapter. Six explosive bolts at the lower bus interface separate the propulsion module from the spacecraft after orbit circularization and spin-down (Figure 1e). The solar-shield boom then is deployed and rigidized with a latching device, as shown in Figure 1f.

The spacecraft bus integrates all engineering equipment for the mission: reaction wheels, cold-gas tank and

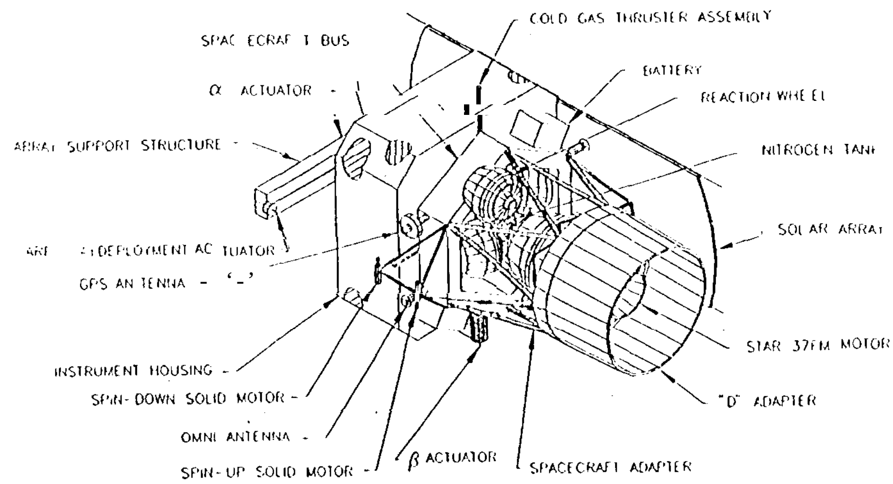

Fig. 1c. Launch configuration, as stowed in the Atlas IIAS shroud.

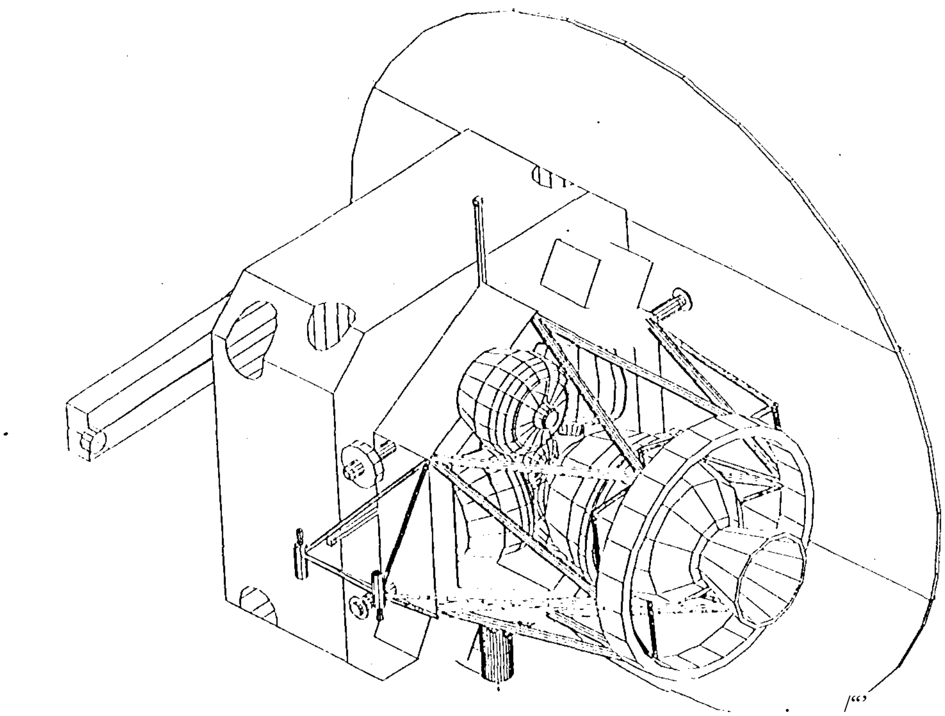

Fig. 1d. Cruise configuration, with solar shield deployed.

thrusters, antennas, batteries, P-actuator attachment, and associated electronics. Its structure and packaging is based on the design used on Voyager and Galileo, and it is sized to provide the same volume as the main bus on Galileo. A hexagonal layout was chosen so that three of the reaction wheels are placed centrally. Battery power is provided by two nickel-hydrogen batteries, which are mounted on opposite sides of the outside of the bus. Omni-directional antennas for telemetry and orbit determination are mounted in pairs on opposite sides of the bus, with pedestals to provide maximum sky coverage. After full deployment, the only obstructions to these antennas will be the solar shield and thruster stalk. Two cold-gas thruster assemblies are incorporated for spacecraft angular-momentum management (see Section 4.4) and the tank for the high-pressure nitrogen is centrally located so that changes to spacecraft inertia properties are minimized as the propellant becomes depleted.

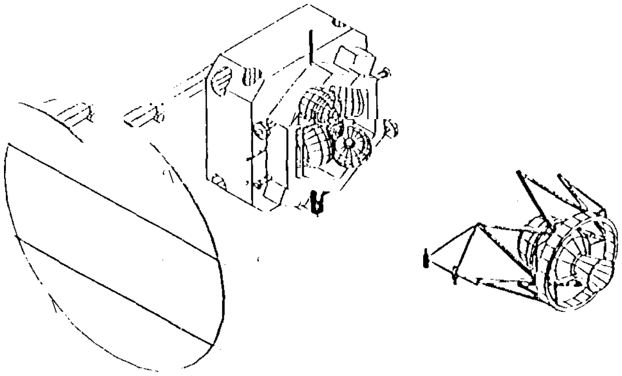

Fig. 1e. POINTS spacecraft just after orbit circularization and spin-down; six explosive bolts at the lower bus interface separate the propulsion module from the spacecraft.

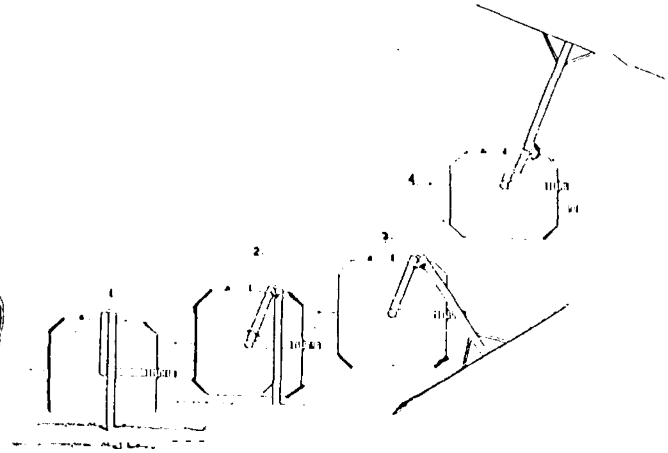

Fig. 1f. Solar-array boom deployment sequence.

The current total mass estimate for the entire spacecraft, including solid rocket motor for orbit circularization, is about 1250 kg, and is itemized in Table 1. About 50% of the mass is in the spacecraft bus, 10% in the instrument housing, and 20% each in the optical benches and the solar shield. This mass leaves a generous margin (~40%) for use of an Atlas IIAS launch vehicle.

3. INSTRUMENT CONFIGURATION

The POINTS instrument comprises two optically identical interferometers placed one above the other, with 2-m baselines whose relative orientation is nominally 90 deg but which can be adjusted by a few degrees in either direction. The wide-angle capability is critical for many of the scientific goals and is advantageous for obtaining rapid closure around the sky and finding bright reference stars. Internal metrology is required for measurement of the angle between baselines and of the starlight optical path differences through each interferometer. Each interferometer comprises two afocal telescopes with parabolic primary mirrors nominally 25 cm in diameter and a magnification ratio of about 10. Further description of the interferometers and their operation can be found in another paper in this volume (Reasenberg, *et al.*).[8]

3.1 Instrument housing

The two interferometers are contained in a box-shaped housing, approximately 1 m high with square cross section roughly 2.6 m on a side. (The corners are cut to fit inside the 3.65-m-diameter circular launch-vehicle fairing.) The housing provides mounting for electronics and tie-down points for cabling, a frame for thermal blankets and heaters, an interface for aperture-cover devices, and protection for sensitive optics during system integration and testing. Thin aluminum sheets (~1.5 mm thick) with stiffeners and local reinforcements would suffice for its construction. Graphite-epoxy panels could be used to reduce mass, but the current mass budget does not indicate any need for such a reduction.

3.2 Optical bench

The optical layout for each interferometer is shown in Figure 2a, and a conceptual design for one of the optical benches is shown in Figure 2b. The graphite-epoxy panel and bulkhead design is based on the optical bench for the Wide Field/Planetary Camera (WF/PC). Graphite-epoxy is an excellent material for precision optical structures because of its high specific stiffness, near-zero coefficient of thermal expansion, and extensive flight history. All components shown are supported by common optics mounting techniques; items with demanding requirements, such as the fiducial blocks, which have critical thermal isolating mounts and heaters (see Section 3,3 below), are contained in separately tested units that have a simple interface to the rest of the optical bench.

One of the optical benches is fixed relative to the spacecraft. The other must be capable of articulation by a few degrees in either direction (from its nominal 90-deg orientation to the fixed bench); see Figure 2c. The fixed bench must provide adequate support to withstand launch loads. The support must be kinematic, to avoid distortion of the bench by the external structure, and there must be thermal isolation between bench and spacecraft. To provide a

Table 1. Mass list for POINTS spacecraft

Component	Mass (kg)
Solar shield:	
18- x 18- cm boom	44
solar shield	81
α actuator	4(I
β actuator	40
actuator electronics (X2)	20
boom deployment device	10
other devices	5
Solar shield subtotal	240
Optical bench:	
bare structure	116
instrument optics & mounts	45
beam-combining optics, detectors, & lasers	22
star trackers (x4)	44
fiducial bloc.ks & thermal control *(x4)*	10
articulation actuator	5
Optical bench subtotal	242
Instrument housing:	
bare structure	91
MLI, thermal control, aperture covers, & mechanisms	41
Instrument housing subtotal	132
Spacecraft 1)11s:	
bare structure	72
thermal control	34
reaction wheels *(x4)*	240
attitude-control electronics ($\times 2$)	20
miscellaneous attitude-control devices	10
propulsion (tank, GN_2, valves, plumbing, thrusters)	28
batteries	74
telecommunications, including antennas	20
GPS equipment, including antennas	6
command & data handling	31
explosive bolts, springs	5
balance ballast	10
cabling	35
power conditioning & distribution	49
Spacecraft bus subtotal	634
'l'()'J'Al,	1248

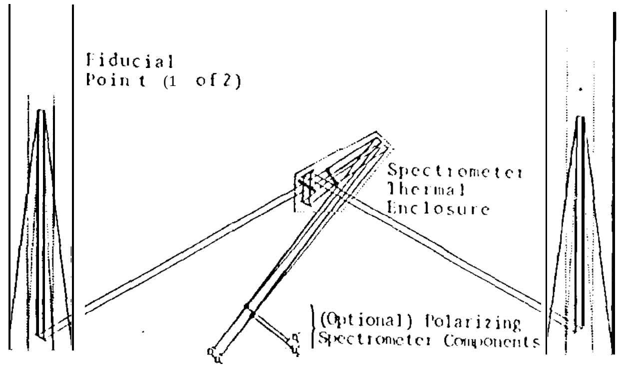

Fig. 2a. Optical layout for each interferometer.

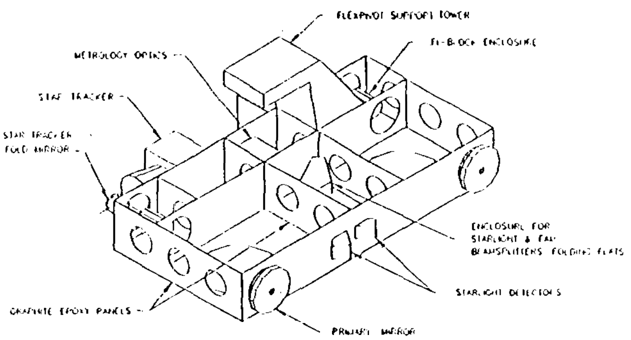

Fig. 2b. Conceptual design for one optical bench (with top panel removed, to show internal layout).

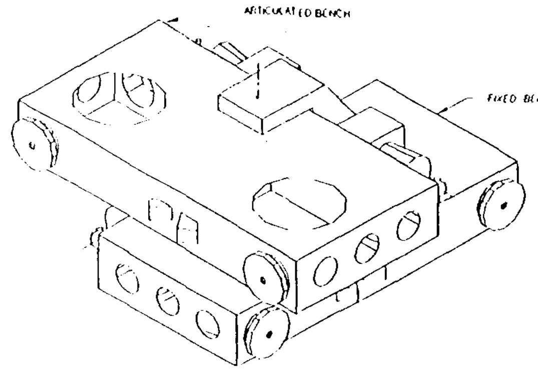

Fig. 2c. Optical bench assembly

structurally complete instrument unit with only one interface to the spacecraft, the optical bench should attach to the spacecraft bus at the same points as the housing or be attached directly to the housing. These mounts might be fiberglass, titanium, or stainless-steel tube trusses or flexures, or fiberglass isolators with aluminum structure elements.

The requirement of a few-degree articulation to be performed on the order of a million times over a 10-year mission suggests use of a flexure rather than a ball bearing, since it can be designed for infinite life and poses no contamination risk. (Ball bearings also suffer from the possibility of developing detents, e.g., caused by jarring during launch.) A "flex pivot" is a commonly used component that combines two or three crossed blades in one convenient unit. Flexpivots sized to withstand launch loads would require only about 24 N (5 lb) linear actuation force per flexure at full travel (for a 1-m moment arm). Flexpivots are generally very stiff and strong in linear degrees of freedom and very soft in rotation; thus, to limit motion to one axis of rotation, they must be used in pairs, with some distance between them along the desired rotation axis. The current design has flexpivots (two or three blades) placed on the top and bottom of the optical bench pair, with a structural support from each bench that reaches around to the opposite side of the other optical bench. (The "flexpivot support tower" is shown in Figures 2b and 2c.) The flight flexpivot system must be designed so that the flexpivots are not kinematically over-constrained: one flexpivot would be isolated from thrust loads (*e. g.*, with a diaphragm flexure), and both must be aligned well enough to avoid excessive moment loads,

3.3 Fiducial block supports

To keep resulting contributions 10 overall measurement error below about 0.2 μas, dimensional changes along critical directions of the four metrology "(fiducial blocks" must be kept smaller than about 2 picometers over time intervals of several hours. Thermal expansion is kept in check by using stable materials such as Ultra Low Expansion glass (ULE®), controlling the average temperature inside the housing to remain near where the glass has the lowest coefficient Of thermal expansion (roughly 20 deg C for ULE®), and limiting temperature variations to a few millidegrees Kelvin through the use of low-conductance supports and special enclosures. (See discussion of thermal control in

Section 5.)

Mechanical stress to the fiducial blocks is limited by using kinematic mounts between each fiducial block and its enclosure, such as the crossed-blade titanium flexures shown in Figures 3a and 3b. Blade segments 5 mm wide, 10 mm long, and 0.5111111 thick were size.d to withstand launch loads. With these mounts and an aluminum enclosure, a mechanical distortion (strain) of the enclosure by about 2 parts per million would result in a 2-pm distortion of the fiducial blocks, due to fore.cs passed across the mounts by the bending stiffness of the blades. In order to keep local distortions at the mount interface from affecting critical dimensions of the fiducial block, the blades arc attac.heel to au Invar disk atop a pedestal of ULE® on either side of the block. The Invar interface exhibits much nigher thermal expansion than the ULE® pedestal, and the pedestal serves to keep the resulting distortions localized.

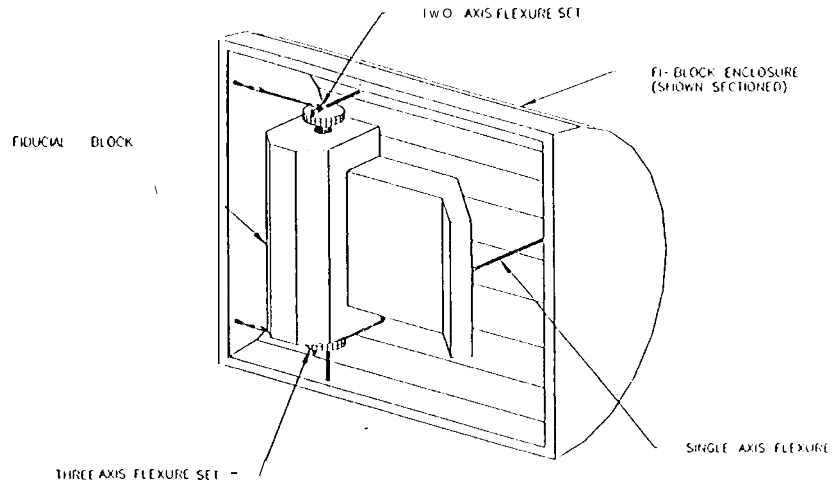

Fig. 3a. Fiducial block support concept

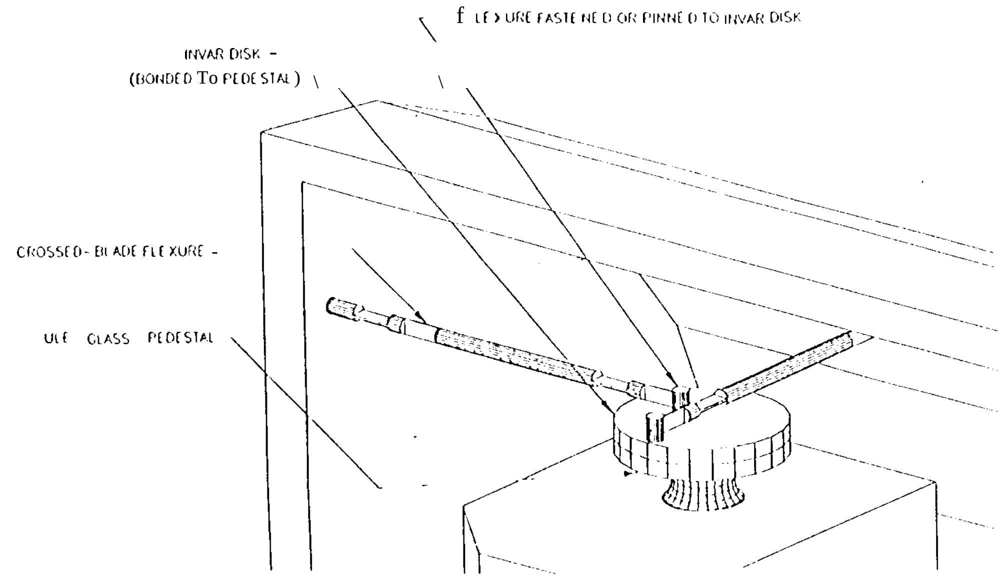

Fig. 3b. Conceptual design of flexure attachment for fiducial block support.

4. ATTITUDE AND ARTICULATION CONTROL SYSTEM

The instrument and spacecraft attitude and articulation control system (AACS) serves a variety of functions, such as maintaining spacecraft attitude during the cruise to apogee, determining spacecraft attitude after orbit circularization, pointing the solar shield to within a few degrees of the Sun line at all times, slewing the spacecraft and instrument for acquisition of target-star pairs, articulating the angle between the two interferometers, providing a target-referenced attitude after each slew to the interferometers for fringe acquisition, maintaining inertial pointing of both interferometers in the presence of pointing disturbances, and managing spacecraft angular momentum. The requirements, strategies, and hardware associated with the most challenging of these functions are summarized below.

4.1 Instrument pointing

As described previously, instrument pointing is accomplished with two single-axis gimbals between the spacecraft and the solar-shield support boom, plus roll of the entire spacecraft around the Sun direction. The angular separation γ_{21} between stars "1" and "2" is estimated from a linear combination of three measurements: measurements by each interferometer ("a" and "b") of the angle between its target star and its optical axis (γ_{1a} and γ_{2b}), and measurement by internal metrology of the angle γ_{ba} between "pseudo baselines" of the two interferometers, which are defined by optics placed at fiducial points along each telescope axis and which are nearly parallel to the actual baselines. Requirements on instrument pointing accuracy, or how closely boresighted the interferometer telescopes must be with the directions to their respective target stars, are determined by the magnitude and dependence on pointing offsets of the difference between the true stellar angular separation γ_{21} and the estimated separation. The lowest-order correction is a linear bias that depends only on the "in-plane" component of the angle between actual baselines and pseudo baselines *i.e.*, on horizontal twisting or yaw within each interferometer. The next largest corrections, which are quadratic in quantities expected to be small, depend only on "out-of-plane" angles target pointing offsets, vertical tilt between the two interferometers, and vertical tilt between the actual and pseudo baselines (see Appendix of Ref. [8] for derivation of the last of these). In a hypothetical worst case, where all these tilts were of comparable magnitude A and of appropriate sign as to add maximally, this quadratic term could be as large as $7\Delta^2$, which would be sub-microarcsecond in magnitude only if A were smaller than about 0.2 arcsec. In practice, these biases can be estimated from a series of measurements, so it is not necessary that pointing accuracy be good enough to make these corrections negligible. The nominal pointing accuracy requirement (2-σ) for each interferometer therefore is taken to be 0.25 arcsec. To maintain high fringe visibility, the pointing must be stable to a few milliarcseconds in-plane and about 0.1 arcsec out of plane (for 95% fringe coherence) over the time needed to measure the fringe intensity as long as 0.1 second for a 15th-magnitude star. For brighter stars, this pointing stability is needed over the nominal "framing" time of about 0.01 s. The current strategy is to use star trackers boresighted with the telescopes to control coarse spacecraft attitude to 0.25 arcsec out-of-plane; however, it may be preferable to mount the trackers on the spacecraft bus instead of the instrument, for convenience and cost savings in integration. The nominal design includes a primary plus one back-up tracker for each interferometer, located as indicated in Figure 2b. Observations are scheduled so that at least one of the interferometers always looks at a star of 10th magnitude or brighter. Fine-pointing in-plane for the other interferometer can be controlled with information from the interferometer looking at the brighter star.

4.2 Instrument slewing

Simulations done at SAO indicate that the average instrument slew required between observations will be about 25 deg; with the Hubble-Space-Telescope (HST) reaction wheels baselined in the nominal configuration, such slews should take less than 60 seconds. This permits achievement of closure around the sky in a few hours, an appropriate recalibration time for estimating system parameters and measurement biases. The time required between observations will include the slew time and the AACS settling time; the latter depends on spacecraft vibrational characteristics and the controller closed-loop bandwidth. For the nominal configuration, with a minimum vibrational frequency of about 2 Hz and modal damping of 0.5%, the time required for structural vibrations to damp to a level consistent with requirements on instrument pointing stability is less than 60 seconds.

Slewing will be accomplished with four HST-type reaction wheels arranged in a pyramidal or tetrahedral configuration (one for redundancy). Information from commercial vendors indicates that available passive isolation technology is sufficient to permit the desired instrument pointing stability noted above. Hence, during observations, the AACS need reject only external torque disturbances; for high Earth orbit, these arise primarily from solar radiation pressure. Spacecraft attitude is estimated from measured body rates provided by an inertial reference unit (IRU), augmented by star-tracker measurements during target acquisition and observation, and possibly by interferometer angle information during each observation. Unmodeled gyro bias, rate scale-factor errors, misalignment errors, and attitude-correction factors are estimated, and then used by a linear proportional-integral-derivative (PID) controller to generate reaction-wheel torque commands. The appropriate controller bandwidth depends on settling time, desired pointing stability, attitude-sensor noise, external torque disturbances (solar radiation pressure), and detector integration time (nominally 0.01 seconds for stars brighter than 12th magnitude). For the nominal configuration, and an assumed sub-arcsecond attitude-estimation error (white noise over a 50-Hz sensor bandwidth), the controller bandwidth will be between 0.01 Hz and 0.15 Hz.

An inertial-grade gyro (drift-rate stability on the order of 0.001 deg/hr) is needed to meet the attitude-control requirement of 0.25 arcsec. This requirement could be met with spinning-mass gyros; however, the desired 10-year mission life suggests the use of a fiber-optic gyro, which also could meet these requirements and whose development is under way for other NASA missions with earlier launch dates.

4.3 Actuator design

Three types of actuators are used for attitude and articulation control of the POINTS spacecraft: the reaction wheels, the solar-shield gimbal actuators, and the interferometer articulation actuator. (For a detailed synopsis of actuator design, see the paper by Agronin[9] in this volume.) Four reaction-wheel assemblies (one for redundancy) are used to control spacecraft motion in three axes and to provide momentum compensation. The limits on structural disturbances during observations can be met with HST-type wheels and slightly modified versions of the HST wheel isolators that are designed to isolate both torque and axial disturbance forms. The HST isolators, which contain a spring and viscous damper in parallel, are configured to take out only axial disturbance forces. Honeywell, Inc. (makers of the HST wheels and isolators) reports that a hexapod isolation system now can be made that isolates torque and force disturbances in all six degrees of freedom.

Two gimbal actuators articulate the solar shield in two axes relative to the spacecraft (the α and β axes shown in Figure 1a). The actuators point the shield toward the Sun, keep the spacecraft in shade, and angle the shield to control angular momentum induced by radiation pressure. These actuators must be capable of 0.1-deg accuracy over a 345-deg (or larger) range of motion, and have a lifetime consistent with spacecraft slews totaling about 0.6 million radians (about 350 slews per day for 10 years, each slew averaging about 25 deg). The baseline design is a standard stepper motor driving a Harmonic-drive gear reducer (200:1) with a high-speed resolver for position sensing (typical accuracy on the order of 3 arcmin). Each gimbal axis has bearings separate from the actuator to support the main loads, the actuator driving the axis through a flexible coupling. Similar actuators have been used to drive solar panels on the Magellan and TOPEX spacecraft.

The interferometer articulation actuator controls the relative orientation between the two interferometers. The orientation is adjusted between observations, and the mechanism is locked during each observation. The actuator, located in the optics housing, is a linear-motor device bridging the two optical benches. The actuator only supports loads between the two interferometers in the direction of articulation; a separate flexure assembly supports and constrains the optical benches in the other five degrees of freedom (see Figures 2b and 2c and Section 3.2). The desired end-point accuracy, as stated previously, is about 0.25 arcsec. Relative in-plane position stability of a few milliarcseconds must be maintained between the two interferometers during an observation.

4.4 Spacecraft angular momentum management

External torques on POINTS will generate angular momentum that initially will be taken up by the spacecraft reaction wheels. A momentum-management system is needed in order to provide control torques to keep the reaction wheels from saturating. Since POINTS will be above the Earth's magnetosphere, momentum management based on magnetic torquers is not feasible. The angle of the solar shield with respect to the Sun could be manipulated so as to use solar-pressure torque as a means of managing momentum; however, the extent to which the shield can be angled is limited by the requirement that the spacecraft remain in shade. (Note that, for certain spacecraft orientations, this limit need not be as severe as that described in section 2.2.) This type of momentum-management system would require no added AACS hardware and could be performed with a relatively simple algorithm. However, this scheme is unable to provide any torque about the line from the spacecraft to the Sun (although this limitation is mitigated partially by the orbit of the spacecraft about the Sun once per year). Furthermore, it could work only during normal operations and is incapable of stabilizing the spacecraft during an emergency. For these reasons, a cold-gas (nitrogen) thruster system has been included for momentum management and contingencies. Initial calculations, based on the expected magnitude of solar-pressure torques and typical leakage rates for such cold-gas systems, show that a supply of about 10 kg of gas should be adequate for all necessary momentum management (including contingencies) during a 10-year mission lifetime.

5. THERMAL CONTROL

Critical metrology optical elements in the interferometers, such as the fiducial blocks and beam-splitter assemblies, will require temperature control at the level of a few millidegrees Kelvin (mK) for periods as long as several hours, in order to prevent thermal-induced motions or distortions from contributing more than about 2 picometers (or 0.2 μas) of error to the measurements of optical path differences. In addition, of course, the materials used for these critical optics must have very low, uniform, stable coefficients of thermal expansion on the order of several parts per billion achievable, for example, with appropriately-sized pieces of premium-grade ULE®. A conceptual design for the temperature-control system for the most critical of these elements - the fiducial blocks - is summarized below.

The back (instrument-facing) surface of the solar shield should have low emissivity in order to minimize radiative thermal coupling between the solar shield and instrument, but not too low, because it is desired to keep the operating temperature for the solar cells under 100 deg C. Representative extremes would be bare aluminum (infrared emissivity on the order of 5%) or black paint (infrared emissivity about 85%), which result in estimated solar-panel temperatures of about 115 deg C and 65 deg C, respectively. The instrument housing is made of thin (~1.5-mm) aluminum sheets, which are covered on the outside with 20-layer Multi-Layer Insulation (MLI).

The inside of the housing would be painted black in order to minimize stray light and help distribute heat loads. Heaters would be distributed around the inside of the housing and used to maintain a uniform ambient temperature of about 20 deg C. An estimated 325 W of heater power would be required, some of which might come from the estimated 250-300 W of power dissipated in the spacecraft bus. Deadband or PID controllers could be used to control the surface temperatures.

The fiducial blocks are made of a material with low coefficient of thermal expansion (better than about 10 parts per billion), about 5 cm in length along the optical axes of the telescopes), and roughly rectangular in cross-section (about 5 cm by 10 cm). To ensure that thermally-induced distortions contribute no more than about 2 pm to the error in optical path measurements, the temperature at each point on each fiducial block must be stable to about 4 mK over a typical recalibration time (several hours). To achieve this, a thermal shield is placed around each fiducial block in the form of an aluminum box or cylinder, whose inside is painted black. In the nominal configuration analyzed, the conservative assumption was made that the shield walls were about 5 mm thick, giving the shield a heat capacity roughly three times that of the fiducial block; thinner walls (~ 1.5 mm) may be feasible. The shield has holes in appropriate places for the metrology laser beams. Closely surrounding each thermal shield is a heater shell. The concept for supporting each fiducial block inside its thermal shield with titanium flexures was described in Section 3.3 and shown in Figures 3a and 3b. In combination with the integral glass pedestals on the fiducial blocks, these flexures provide the near-kinematic support needed to minimize mechanically-induced stresses: they also are a source of heat conduction to the fiducial blocks. Each thermal shield could be mounted to the optical bench with graphite-epoxy blades in a "spider" configuration similar to that used in telescopes for secondary mirror mounts. The heater shell would be mounted either to the thermal shield or directly to the optical bench in such a way as to keep structural loads as continuous as possible, while still providing relatively high thermal resistance between the shell and the thermal shield. This high thermal resistance gives higher leverage to the heater control. Heat transport between the heater shell and the housing is much greater *via* radiation than *via* conduction. Although further thermal isolation of the fiducial block is not indicated to be necessary by analyses thus far, it could be achieved by splitting fiberglass tube segments into the titanium rod sections. For the ultimate in thermal isolation, a fiberglass band system could be designed.

With the above assumptions, analyses using detailed models for individual fiducial blocks, thermal shields, and heater shells, and simpler models for the instrument housing and solar array, gave the following results. First, with no active control of the housing temperature, the worst-case steady-state temperature gradient across the interior of the instrument housing was found to range from 14 K to 1.7 K, corresponding to the back of the solar shield being black or bare aluminum, respectively. This worst-case condition would occur if the housing were oriented so that one of its square sides remained in a fixed position, parallel to the solar shield and facing space, for as long as 24 hours. The worst-case housing gradient of 14 K produced a gradient across the fiducial block that was smaller than 1 mK. Alternatively, a steady-state temperature gradient as large as 63 K would have to exist across the interior of the housing to produce a 2-mK gradient across the fiducial block. Second, two types of transient conditions were considered: a maximum-duration Earth occultation, lasting 1.8 hours, and a 180-deg instrument slew lasting 10 minutes. (Other possible transient conditions, such as those caused by on-board thermal disturbances, would have much smaller effects.) The instrument slew caused the greater rate of change in thermal gradient across the instrument housing - on the order of 3.7 K/hr if the back of the solar shield is painted black or white. The corresponding rate of change for the temperature gradient across the fiducial block would be no larger than 0.1 mK/hr. Thus, this thermal control scheme appears sufficient for maintaining thermal stability of the fiducial blocks to within the desired several mK over observation intervals of several hours. Similar, though probably less complicated, control schemes would be used for other critical optical elements, such as the beam-splitter assemblies.

6. SYSTEMS ISSUES

6.1 Launch sequence

The baseline selection for the POINTS launch vehicle is General Dynamics' Atlas IIAS with a Centaur upper stage and a Star 37FM (Thiokol) solid-rocket motor for orbit circularization. This combination is capable of delivering a payload of approximately 1750 kg to a 100,000-km-radius" circular orbit. (This includes a conservative deduction of 7% of the Atlas-Centaur capability for various contingencies, in addition to the 99%-confidence Flight Performance Reserve.) The payload capability is only a slowly varying function of the orbit altitude. The current best estimate for the POINTS payload is about 1250 kg (see Table 1), which leaves a generous (~ 40%) design margin.

The initial launch of POINTS will be into a transfer orbit with an altitude of approximately 167 km, after which the Centaur upper stage will be fired to insert POINTS into an elliptical transfer orbit with an apogee of approximately 100,000 km. Following separation from the Centaur, the solar shield will be deployed partially in order to provide a required power of approximately 340 Watts during the 17-hr cruise to the 100,000-km radius; the spacecraft will be 3-axis-stabilized during this long cruise. Just prior to ignition of the Star 37FM for orbit circularization, a small spin-up solid motor will be fired to provide a spin rate of a few RPM for the 60-second circularization burn, and a similar motor will be fired following the circularization to de-spin the spacecraft, which then will be returned to its 3-axis stabilized mode. Following the circularization, the propulsion module will be jettisoned, and deployment of the solar shield will be completed. Given the magnitudes of the expected errors in both the orbit-injection and circularization burns, the perigee and apogee should be within 5% of the nominal values, completely within the acceptable range for the POINTS mission.

The inclination of the orbit with respect to the ecliptic will influence the occultation history of the spacecraft; higher inclinations will reduce the lengths of the occultation seasons. Inclination with respect to the equator is determined only by launch site and direction, while inclination with respect to the ecliptic also depends on launch date and time. The inclination to the ecliptic can be changed at launch, at transfer-orbit insertion, or at the time of orbit circularization; a change requires the least energy if performed at the time of orbit circularization. At that time, a change in inclination of 15 deg would cause a penalty of about 25 kg in delivered payload, while a change of 25 deg would lead to a penalty of about 65 kg.

6.2 Power requirements

The current estimate for the POINTS power consumption is a peak power of 670 Watts and an average power of 600 Watts, including a 25% contingency in each case. A 4.8-m-diameter circular solar panel with silicon cells and a low-emittance rear surface would provide an operating temperature of 100 deg C, at which temperature the array could provide 800 Watts after 10 years of operation. (Gallium-arsenide cells could provide nearly twice as much power but do not appear to be required.) Thus, the size of the solar shield is determined by requirements on shading the spacecraft and target accessibility, rather than power requirements. Two NiH batteries providing a combined capacity of 3200 Watt-hr have been assumed; a discharge of less than 40% would be required for full operation even during the maximum Sun-occultation time of approximately 1.8 hr. In emergencies that cause the spacecraft to go into a safing mode and search for the Sun, full-power operation of the attitude-control, telemetry, and computing subsystems could be maintained for as long as ten hours without solar power, with operation at a lower power possible for a much longer period.

6.3 Spacecraft velocity determination

The right-angle orientation of the two POINTS interferometers accentuates the need to correct astrometric measurements for the effect of stellar aberration, which produces an error in angular measurements that scales linearly with the spacecraft velocity and the sine of the angle between stars. In order for this error to be smaller than 1.0 μas, the POINTS spacecraft velocity must be determined to better than 1.5 mm/s; in fact, the strategy described below is capable of determining the spacecraft velocity to better than 0.6 mm/s (0.4-μas error contribution). In high Earth orbit, the primary external source of disturbance to the spacecraft acceleration is solar radiation pressure, the effects of which can be modeled quite accurately for the nominal POINTS configuration. Effects from atmospheric drag and anomalies in Earth's gravitational field, which dominate the unmodeled spacecraft acceleration in low orbits, are negligible. Calculations show that the typical rate of leakage from the cold-gas system also will give small accelerations compared to the uncertainties in the acceleration due to solar radiation pressure.

At a radius of 100,000 km, POINTS would be well outside the constellation of Global Positioning System (GPS) satellites (~27,000 km orbital radius). Although this makes it impractical to determine velocity by using an on-board GPS receiver, the alternative of having POINTS mimic a GPS satellite is predicted to meet the velocity requirements.[10] POINTS would carry a GPS-like beacon having multiple tones in the frequency ranges of 1.2-1.6 GHz (L band) or near 15 GHz (Ku band); the multiple tones are used to correct for ionospheric effects. The beacon would be tracked by eight-channel GPS receivers located around the world, with each receiver simultaneously tracking seven GPS satellites and POINTS. Using only the six TOPEX sites and tracking for only two hours out of every eight, the predicted velocity error for POINTS would be about 0.1 mm/s after a single four-day orbit. Even after another six days without tracking data, the velocity uncertainty would grow to only 0.25 mm/s, still meeting the requirements. Thus, it is likely that the POINTS velocity requirements could be met with a tracking duty cycle under 10%.

The spacecraft and ground hardware required for the orbit determination would be simple. The spacecraft would broadcast a low-power signal (broadcast power of approximately 1.5 Watts) through one of a pair of switched omni-directional (3 dB) antennas. Since omni-directional antennas could be used, there would be no need to interrupt the scientific observing to point these antennas at the Earth. Occasional interruptions in the beacon signal would be required in order to change from one antenna to the other, but this is of little consequence given the low requirement for the overall tracking duty cycle. Simple modeling of the antenna position relative to the spacecraft center of mass should suffice to reduce all beacon data to the proper reference frame. An ultra-stable oscillator with frequency stability $\delta f/f \approx 7 \times 10^{-11}$ over one second (or 7×10^{-12} at Ku band) is required on board to enable a coherent integration time of about one second, which should be adequate to detect the beacon signal. The total mass of the spacecraft hardware would be about 5 kg, with a total power consumption of 20 Watts. If the standard GPS L-band channel were employed, off-the-shelf GPS receivers with omni-directional antennas could be used on the ground. Use of Ku band would require provision of ground antennas with about 20 dB of gain (corresponding to beamwidths on the order of 15 deg, or antenna diameters on the order of 10 cm) that would provide the input to one channel of each GPS receiver, a relatively simple and inexpensive addition to each ground observing station. In this case, it would be necessary to estimate the delay between the Ku- and L-band signals in the ground receivers, which may be the largest contributor to the velocity error quoted above.

6.4 Telemetry and data storage

POINTS will have two detector arrays for each of the two interferometers. The current proposal is that each detector be divided into 650 optical frequency bands. If the photon count from each band is accumulated for one second and stored as a 16-bit number, the interferometer data rate would be approximately 42 kilobit/s, with a total data rate of about 50 kilobit/s after adding 20% to account for metrology data and other overhead. The spacecraft science data would be downlinked daily for an hour or less. Assuming 15 hours per day of actual observing and a one-hour downlink, the required telemetry rate (with no data compression) would be 750 kilobit/s.

An S-band (2.3-GHz) downlink from an omni-directional antenna is assumed. Two switched antennas, located on opposite sides of the spacecraft, would be used. If the downlink occurs during an observation sequence, it would be necessary to switch between the two antennas occasionally, increasing the total time required for the downlink by up to a few minutes per day. Alternatively, since only one hour of each day would be required for the data downlink, it also would be possible to stop observations to perform both the uplink and downlink as well as to acquire a substantial quantity of orbit-determination data.

The ground receiving antennas for the telemetry would be the standard Deep Space Network (DSN) 26-m antennas used for high Earth orbiters; these antennas are located in California, Spain and Australia. For a 2-dB loss in signal-to-noise ratio relative to that for a clear, dry atmosphere, a transmitted power of 7.5 Watts would be adequate to provide a bit error rate less than 10^{-4}, even with simple quadrature-phase-shift-keying (QPSK) modulation. (Weather severe enough to cause a loss of 2 dB or more should occur less than 5% of the time.) With the same transmitted power and a 2.3- or 8-GHz downlink to the subnet of 11-m antennas currently being constructed by the DSN, a phased array or small fixed antenna would be required with a beamwidth on the order of 30 deg (or larger, if more sophisticated data compression and modulation schemes were used). A maximum of one hour per day would be required for the data downlink; if a phased array were placed on the outside of the instrument housing opposite the spacecraft bus, the downlink requirement might be reduced to several minutes per day. The high Earth orbit offers a long period of visibility for each ground station; this provides flexibility in scheduling the downlink, reduces conflicts with other missions, minimizes the overhead time required for ground-station setup, and avoids operational problems associated with a need for multiple downlink sessions each day over a 5- to 10-year mission.

With the above scenario for telemetry transmission, at least one day of science data must be recorded on the spacecraft. At a data rate of 50 kilobit/s for 15 hr, the total quantity of data accumulated in a single day would be about 340 megabytes. Thus, the baseline of 500 megabytes of solid-state memory that has been selected for the Cassini mission also would be a good choice for POINTS. Storage capability could be improved by increasing the amount of solid-state memory or planning some on-board data compression.

7. ACKNOWLEDGMENTS

We gratefully acknowledge the technical contributions of C. Dunn, B. Haines, S. Lichten, C. Peng, G. Siebes, D. Spencer, M. Vincent, and L. Young to the work described here. R. Bartman, C. Briggs, R. Laskin, and G. Sevaston of JPL also have supplied useful guidance and support, for which we thank them. We also are grateful to the members of SAO's POINTS team, led by R. Reasenberg; their comments and interactions have been extremely useful throughout the course of the JPL studies. This work has been performed by the Jet Propulsion Laboratory, California Institute of Technology, under contract with the National Aeronautics and Space Administration.

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